

NASA DOUBLE ASTEROID REDIRECTION TEST (DART) TRAJECTORY VALIDATION AND ROBUSTNESS

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The Double Asteroid Redirection Test (DART) mission will be the first to test the concept of a kinetic impactor. Several studies have been made on asteroid redirection and impact mitigation, however, to this date no mission tested the proposed concepts. An impact study on a representative body allows the measurement of the effects on the target's orbit and physical structure. With this goal, DART's objective is to verify the effectiveness of the kinetic impact concept for planetary defense. The spacecraft uses solar electric propulsion to escape Earth, fly by (138971) 2001 CB₂₁ for impact rehearsal, and impact Didymos-B, the secondary body of the binary (65803) Didymos system. This work focuses on the heliocentric transfer design part of the mission with the validation of the baseline trajectory, performance comparison to other mission objectives, and assessment of the baseline robustness to missed thrust events. Results show a good performance of the selected trajectory for different mission objectives: latest possible escape date, maximum kinetic energy on impact, shortest possible time of flight, and use of an Earth swing-by. The baseline trajectory was shown to be robust to a missed thrust with 1% of fuel margin being enough to recover the mission for failures of more than 14 days.

INTRODUCTION

Planetary defense is gaining more and more attention over the years as our awareness of the space environment expands and the risk of small and medium impacts increase. Events like the Chelyabinsk meteor¹ and the Tunguska impact,² among many others, serve as a reminder of the importance of planetary defense research and efforts. The majority of potentially hazardous asteroids (PHAs) lie in the range of 50 to 200 km in diameter³ with about 5000 objects found to date. Different mitigation techniques have been studied and can be available for immediate use, the most promising technologies make use of energetic explosion, gravity tractor, kinetic impactor, or directed energy. From those options, the kinetic impactor is effective for a wide range of the warning times and is more effective for objects with the diameter range of the majority of PHAs, Fig. 1.

In line with the global efforts in planetary defense, different space missions begin to study, test, and prepare the necessary structure for an eventual redirection of a hazardous object, e.g. AIDA, Dawn, NEOWISE, OSIRIS-Rex, Hayabusa 1 and 2, etc. The Double Asteroid Redirection Test (DART) aims to be the first mission to test the concept of an asteroid kinetic impactor for planetary defense.^{4,5} The mission's target is Didymos-B, here named Didymoon, the secondary body of the

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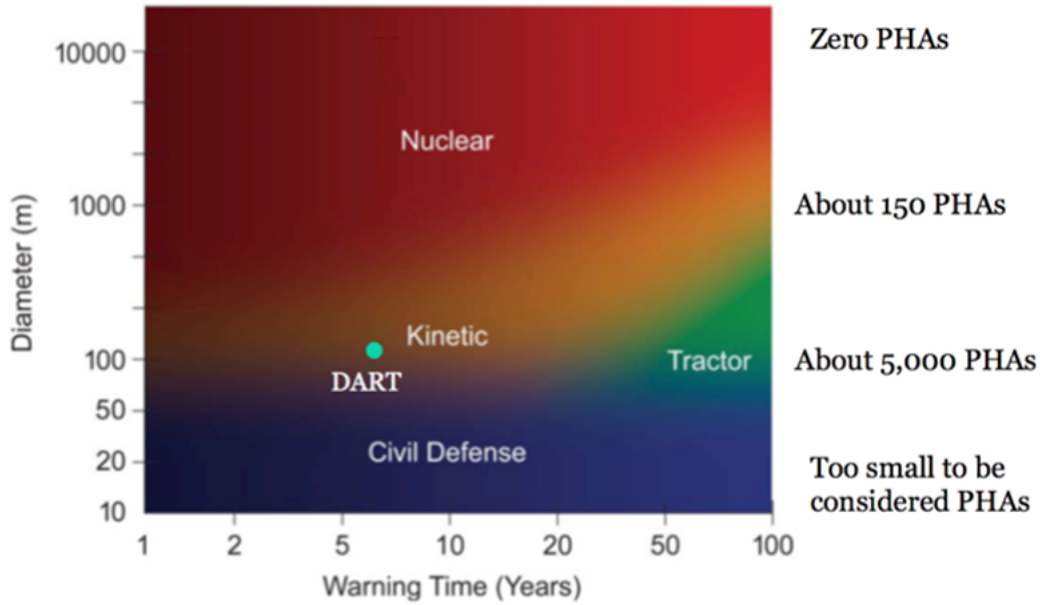


Figure 1. The four types of mitigation and their regimes of primary applicability³

near-Earth, Apollo-type binary-asteroid system (65803) Didymos.⁶ For the purpose of this study, the main body of the system, Didymos-A, is named Didymain. DART's objective is to impact Didymoon to change its orbit and allow the characterization and measurement of the deflection. The spacecraft is a medium class 638 kg with the latest technology in solar electric propulsion (SEP): NASA Evolutionary Xenon Thruster (NEXT).⁸ DART's trajectory is divided into two distinct phases: Earth escape through a powered spiral, and low-thrust heliocentric transfer. During its transition from Earth to the Didymos system, DART will fly by (138971) 2001 CB₂₁.⁷ This intermediate flyby is strategic for the mission final operations, because it allows sensor calibration and control-gain tuning prior to the impact. To maximize the deflection measured from Earth, the mission is constrained by:

- Impact date, to prioritize ground based radar and optical observation;
- Solar phase angle at impact, for the terminal optical guidance system; and
- Impact angle, to maximize the measurable effect in Didymoon's orbit.

Being a precise targeting mission with limited propellant available for maneuvering, a second important characteristic of the main trajectory is the need for robustness to missed thrust events.

This paper assesses the mission's heliocentric considering systems requirements and spacecraft constraints. The analysis consists in the development of a heliocentric trajectory baseline that complies with all mission constraints. Mission characteristics and constraints are all incorporated into the optimization tool, which eliminates the need for post processing the results. Once the baseline is obtained, it is assessed against different performance indexes. To understand the baseline trajectory's performance, trajectories with different objective functions are optimized and compared.

And, its robustness against missed thrust events is verified by introducing a new technique based in an objective function that allows the spacecraft to coast as much as possible before re-starting the engines.

Next section **Methods** presents an overview of the mission with its main characteristics and the rational used in performing the studies. Section **Mission Constraints** outlines the modeling of the constraints used in the trajectory design. Sections **Baseline Trajectory and Performance** and **Missed Thrust Analysis** in the **Trajectory Analysis** show, respectively, the baseline trajectory design and performance against other mission objectives and missed thrust events. Section **Alternative Scenarios** considers the results in the previous two sections to design contingency trajectories. Finally, **Conclusions** presents the summary of this work.

METHODOLOGY

The calculation and design of the Earth escape spiral takes into account critical subsystem requirements, such as maximum time in the radiation belt and eclipses, programmatic launch window, communication and power requirements. The evaluation of the Earth spiral phase is out of the scope of this study, but its final state vector is the initial condition for the Heliocentric phase. The Heliocentric transfer design includes a 30-day forced coast prior to the midcourse flyby and impact for target identification, trajectory correction maneuvers, and close approach autonomous navigation. The overall trajectory design was changed once the mission was changed from impulsive thrust to low-thrust propulsion. The change had three main positive impacts on the overall mission. First, the mission is cheaper. A launch vehicle cost reduction was possible with the capability of using SEP to escape Earth. The launch can be made as a rideshare on families of a geostationary transfer orbits with the possibility of using different commercial launchers. Second, the mission is more flexible. A midcourse flyby is assured and a larger launch window was obtained. Third, the trajectory is more robust to impact conditions and misses, as well as missed thrust.

DART's baseline Heliocentric transfer is optimized for minimum propellant consumption considering the escape conditions, midcourse flyby and impact targets with forced pre-coast, and impact constraints. The SEP system uses a fixed flow rate and single operating thrust point for simplicity with a throttle level of 28 (TL28 - from NEXT Throttle Table 11⁸), and a fixed specific impulse, Isp, end-of-life value for design conservatism. The NEXT engine has nearly 38 kg of Xenon available for the Heliocentric phase and its performance values, used in this study, represent the most conservative scenario: fix specific Isp of 3093.03 s, maximum thrust of 0.137121 N, and 90% duty-cycle. Table 1 outlines the mission mass budget and Table 2 summarizes the spacecraft and mission characteristics. The trajectory optimization is made through the Evolutionary Mission Trajectory Generator (EMTG) tool: a NASA Goddard Space Flight Center trajectory design software.^{9,10} The constraint in the impact angle has its in-plane component evolve faster than the other variables, which makes the convergence difficult and creates several local minima. EMTG's monotonic basin hopping¹¹ feature allows a particularly smooth convergence despite this uneven evolution and improves the search for a global minimum. Once the baseline Heliocentric transfer is designed, a second analysis is performed to evaluate the performance of this trajectory with respect to other mission objectives: latest escape possible, maximum energy at impact, variation of escape and impact dates, and the use of an Earth swing-by. Finally, the main trajectory is also analyzed for robustness against engine failure. The missed thrust event evaluation takes into consideration the maximum coast time allowed throughout its path considering different margins for Xenon and a 5% increase in duty cycle in three scenarios:

1. Reach the exact same impact conditions;
2. Reach a solution that complies with all constraints, but not necessary equal to the main trajectory; and
3. Reach a solution that complies with relaxed constraints.

Table 1. DART spacecraft mass allocation

Component	[kg]
DART MEV dry mass	483.0
Margined hydrazine	26.5
Neutral mass	509.5
Deterministic Xenon propellant	116.0
Operational xenon margin	3.0% det. 3.5
Missed thrust Xenon margin	5.0% det. 5.7
Xenon residuals	1.0% total 3.0
Delivered mass	530.0
Total Xenon	128.5
Total wet mass	638.0

Table 2. DART Spacecraft and Mission Characteristics

Earliest escape date (zero energy state)	October 3rd, 2021
Ion engine Thrust	0.137121 N
Ion engine Isp	3093.03 s
Duty-cycle	0.9
S/C escape mass	568.105 kg
S/C neutral mass	≥ 519.00 kg

MISSION CONSTRAINTS

The spacecraft escapes Earth with a characteristic energy, C_3 , close to zero and due to its modest propulsion capacity, it keeps an orbit close to 1 A.U., which satisfies most of the thermal and power constraints. The driving trajectory requirements will come from the pre-flyby forced coast, pre-impact forced coast and impact constraints. The spacecraft requires 30 days coast prior to the impact to identify the target, perform trajectory correction maneuvers, and impact the asteroid. The pre-flyby or impact rehearsal also uses a 30-day force coast to calibrate the spacecraft sensors and tuning the control gain which will be used to autonomously drive the spacecraft to impact. The constraints on the Didymos system arrival and Didymoon impact are: impact date, solar phase angle and impact angle. The next subsections explain each constraint in detail.

Impact Date

DART is intended to impact the Didymos system at a time near its conjunction with Earth. To this end, the impact date, t_{impact} , is constrained to be between September 25 and October 20, 2022.

$$\text{Sep. 25, 2022} \leq t_{\text{impact}} \leq \text{Oct. 20, 2022} \quad (1)$$

Solar Phase Angle

DART's terminal guidance operates using images from an optical telescope. The Didymos target scene is illuminated by the Sun only. Lighting affects detection range as well as centroiding accuracy. In addition, the lighting conditions have implications for the ability to reconstruct the impact point using the final downlinked images. The solar phase angle, ϕ_S , is the angle connecting the instantaneous spacecraft-Didymoon-Sun points. Since it is difficult to use the spacecraft position relative to Didymoon at the time of impact (they are identical), the relative velocity can be used. For mission success, the arrival solar phase angle must be less than 60° .

$$\phi_S = \cos^{-1} (\hat{\mathbf{r}}_{Sun/D2} \cdot -\hat{\mathbf{v}}_{sc/D2}) \quad (2)$$

$$\phi_S \leq 60^\circ \quad (3)$$

Where, $\hat{\mathbf{r}}_{Sun/D2}$ is the unit vector pointing from Didymoon to the Sun and $\hat{\mathbf{v}}_{sc/D2}$ is the unit vector associated with the velocity of the DART spacecraft relative to Didymoon.

Impact Angle

The asteroid impact deflection experiment is most observable if DART imparts momentum/energy into Didymoon's orbit semimajor axis (relative to Didymain). This goal places constraints on the orientation of DART's arrival velocity with respect to Didymoon's orbit velocity about Didymain. The impact angle is defined as the angle between the spacecraft arrival velocity at Didymoon, $\mathbf{v}_{sc/D2}$, and Didymoon's velocity relative to Didymain, $\mathbf{v}_{D2/D1}$.

$$\phi_I = \cos^{-1} (\hat{\mathbf{v}}_{sc/D2} \cdot \hat{\mathbf{v}}_{D2/D1}) \quad (4)$$

This angle can be deconstructed into two components, an in-plane angle and an out-of-plane angle. These are relevant because momentum/energy that is imparted out-of-plane changes Didymoon's orbit plane, which is much less observable than in-plane changes to orbit period. To this end, DART's arrival relative velocity must lie near to Didymoon's orbit plane. The two angles are computed and constrained as follows.

Out-of-Plane Impact Angle. The Out-of-Plane Impact Plane Angle, ϕ_{OP} , is a signed angle that must lie between $\pm 30^\circ$. The sign indicates the direction of the angle, where $+90^\circ$ points opposite to Didymoon's relative orbit angular momentum vector, 0° is in the Didymoon relative orbit plane, and -90° is directed parallel to Didymoon's orbit angular momentum vector. For all practical purposes, the orbit angular momentum vector for Didymoon is considered constant over the simulation and does not need to be continuously recomputed. The out-of-plane angle is therefore most sensitive to the incoming spacecraft velocity.

$$\phi_{OP} = \cos^{-1} (\hat{\mathbf{v}}_{sc/D2} \cdot \hat{\mathbf{h}}_{D2/D1}) - 90^\circ \quad (5)$$

$$\mathbf{h}_{D2/D1} = \mathbf{r}_{D2/D1} \times \mathbf{v}_{D2/D1} \quad (6)$$

$$-30^\circ \leq \phi_{OP} \leq 30^\circ \quad (7)$$

Where, $\hat{\mathbf{v}}_{sc/D2}$ is the unit vector associated with the instantaneous velocity of the DART spacecraft relative to Didymoon, $\hat{\mathbf{h}}_{D2/D1}$ is the unit vector pointing along the orbit angular momentum of Didymoon relative to Didymain, $\mathbf{r}_{D2/D1}$ is the instantaneous position of Didymoon relative to Didymain, and $\mathbf{v}_{D2/D1}$ is the instantaneous velocity of Didymoon relative to Didymain.

In-Plane Impact Angle. The In-Plane Impact Angle is an unsigned angle that relates the orientation of the DART arrival velocity with Didymoon's instantaneous velocity about Didymain, projected into Didymoon's orbit plane. This angle is meant to be either close to 0 or 180° in order to maximize the effectiveness of the mission experiment. For the current DART and Didymos system geometry, the desired angle is 180°, because this orientation places the impact on the sunward side of Didymain, which improves local lighting conditions. The angle is computed by constructing a local coordinate system that is aligned with Didymoon's orbit angular momentum. For all practical purposes, this coordinate system is fixed over the simulation. However, the constrained in-plane angle depends on Didymoon's velocity relative to Didymain, which is changing with a period of roughly 11.9 hours. This makes the angle most sensitive to small (minute or hour) changes in arrival time.

The local coordinate system is constructed using the Didymoon angular momentum vector ($\hat{\mathbf{z}}^A = \hat{\mathbf{h}}_{D2/D1}$) and an arbitrary reference vector, $\hat{\mathbf{y}}_{ref}$. The superscript "A" denotes this arbitrary coordinate system and superscript "I" denotes the inertial coordinate system that the inputs are provided in.

$$\mathbf{Q}^{A/I} = \begin{bmatrix} \hat{\mathbf{x}}^A & \hat{\mathbf{y}}^A & \hat{\mathbf{z}}^A \end{bmatrix} = \begin{bmatrix} \frac{\hat{\mathbf{y}}_{ref} \times \hat{\mathbf{z}}^A}{|\hat{\mathbf{y}}_{ref} \times \hat{\mathbf{z}}^A|} & \frac{\hat{\mathbf{z}}^A \times \hat{\mathbf{x}}^A}{|\hat{\mathbf{z}}^A \times \hat{\mathbf{x}}^A|} & \hat{\mathbf{h}}_{D2/D1} \end{bmatrix} \quad (8)$$

The in-plane components of these vectors can be specified by nulling out the bottom row of this rotation matrix.

$$\mathbf{Q}^{A/I} = \begin{bmatrix} Q_{11} & Q_{12} & Q_{13} \\ Q_{21} & Q_{22} & Q_{23} \\ 0 & 0 & 0 \end{bmatrix} \quad (9)$$

$$\hat{\mathbf{v}}_{D2/D1}^A = \mathbf{Q}^{A/I} \hat{\mathbf{v}}_{D2/D1}^I \quad (10)$$

$$\hat{\mathbf{v}}_{sc/D2}^A = \mathbf{Q}^{A/I} \hat{\mathbf{v}}_{sc/D2}^I \quad (11)$$

$$\phi_{IP} = \cos^{-1} \left(\hat{\mathbf{v}}_{sc/D2}^A \cdot \hat{\mathbf{v}}_{D2/D1}^A \right) \quad (12)$$

$$175^\circ \leq \phi_{IP} \leq 180^\circ \quad (13)$$

Where, $\mathbf{Q}^{A/I}$ is the constructed constant rotation matrix that maps the inertial inputs into the in-plane coordinate system.

TRAJECTORY ANALYSIS

To comply with DART's main mission objective to successfully impact Didymoon in a way that generates a measurable change in its orbit, the Heliocentric transfer analysis has three objectives: generate a baseline trajectory that complies with all problem constraints, compare the performance of the baseline trajectory against other mission priorities, and assess the spacecraft's recoverability in face of missed thrust. All the design results have to take into account the aforementioned constraints.

Baseline Trajectory and Performance

Case (1): Maximum Final Mass - Baseline. As mentioned before, the baseline includes a mid-course flyby of (138971) 2001 CB₂₁ for impact rehearsal and a 30-day forced coast prior to both encounters. The trajectory is optimized to deliver the maximum spacecraft final mass, this objective serves as a metric to define a preliminary the size of the spacecraft and a range for its mass budget. The Earth escape date for this case comes from the mission development program and is set to October 3, 2021. The resulting baseline Heliocentric transfer, Fig. 2, has a thrust-coast-thrust structure that makes it flexible to impact condition changes. Although the low-thrust control structure is similar to what is usually seen in a simple rendezvous case, this trajectory is essentially a flyby type final condition. Each thrust arc acts to control a different portion of the trajectory, the first thrust arc is almost entirely dedicated to provide energy to increase the orbit's velocity, it adjusts for the correct midcourse flyby inclination, and places the orbit into a close resonance to Didymos' orbit. The first arc also adjust for the solar phase angle and the out-of-plane component of the impact angle. The second arc corrects the velocity vector, and fine tunes the impact date together with the impact angle's in-plane component.

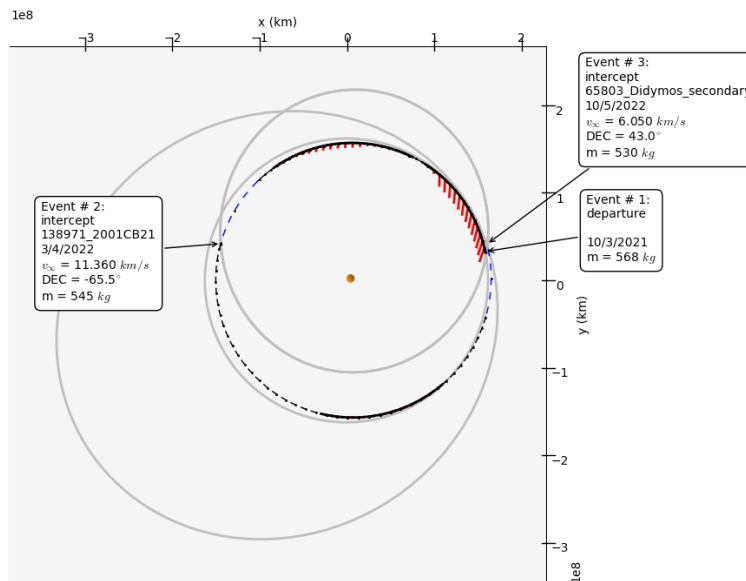


Figure 2. Case (1): DART baseline Heliocentric transfer

Case (2): Escape Earth as late as possible. Delay on the launch can happen for many reasons and these delays will carry to the escape date. Therefore, the next analysis studies the latest possible escape date that generates a feasible trajectory. The delivered mass found on case (1) is used as

the minimum dry mass for this optimization to guarantee that the same spacecraft can achieve the mission in case of delays. Figure 3 shows the resulting trajectory, note that it is possible to delay 73 days on the escape and still complete the mission delivering 530 kg with a similar thrust structure.

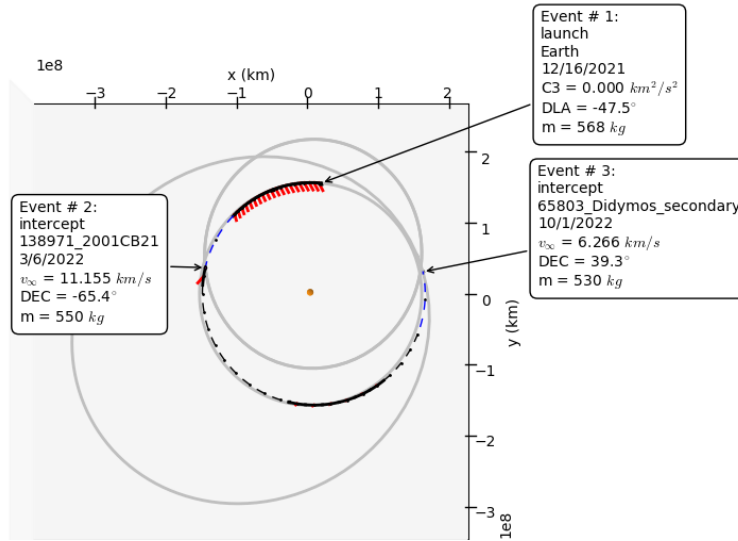


Figure 3. Case (2): Optimal trajectory that escapes Earth as late as possible

Case (3): Maximum kinetic energy on impact. A good metric for the baseline is to measure how much extra energy could be delivered at impact. This solution optimizes the maximum kinetic energy delivered to the system. The same minimum dry mass of 530 kg is used with an unconstrained escape date. The solution, Fig. 4, results in 1709 kJ imparted at impact compared to the 1603 kJ from the baseline. The gain of 103 kJ is small and does not change the order of magnitude of the change in Didymoon’s orbit. This result shows that the baseline is performing well with respect to the maximum possible delivered energy metric.

Case (4): Earlier impact date, and Case (5): Earlier escape and impact date. Both optimization problems are solved with the objective of minimizing the time of flight. This metric helps to understand how fast the mission can be made considering an unconstrained impact date in case (4) and unconstrained escape and impact dates in case (5). Figures 5 and 6 show, that it is possible to gain 4 hrs, case (4), and 79 days, case (5), from the baseline. However case (5) gains only 3 day when compared to case (2). The results outline the importance that the solar phase constraint and minimum dry mass have in the trajectory design, both drove the solutions to similar results as the baseline in case (4) and the latest Earth escape on case (5). As a result, cases (1) and (2) perform well when compared with the minimum time of flight metric.

Case (6): Benefits of an Earth swing-by. As mentioned before, the low C_3 combined with low thrust produces a trajectory that has its radius close to 1 A.U.. This distance suggest that an Earth gravity assist could be used to change the trajectory’s velocity vector. Propellant can be potentially saved with a swing-by by having the control targeting the Earth and adjusting the spacecraft’s velocity magnitude appropriately. The swing-by would then be responsible for directing the velocity vector to the correct Didymoon impact alignment. For the DART case, the result of this type of trajectory, Fig. 7, consumes 5.5 kg more propellant to target Earth and re-target Didymoon. Therefore, an Earth gravity assist will not improve the trajectory; once more, the baseline performs well.

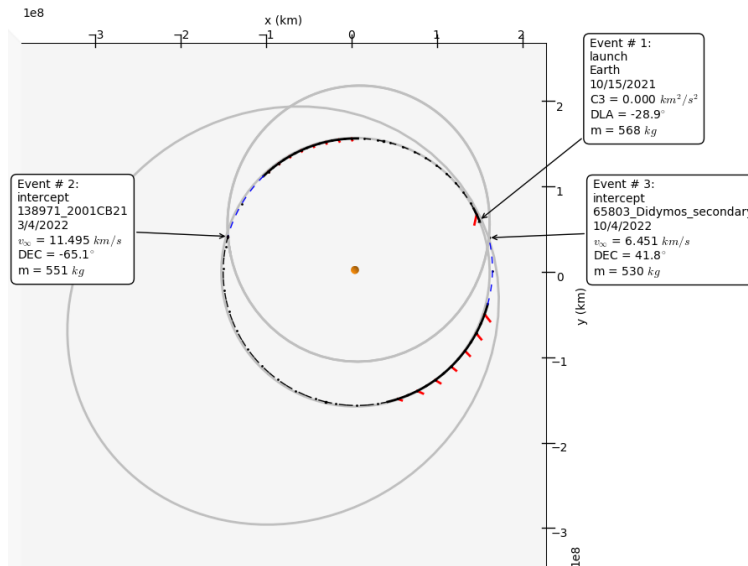


Figure 4. Case (3): Optimal trajectory that delivers the maximum kinetic energy on impact

Summary of Performance Results. Table 3 presents a summarized comparison of the above solutions against the baseline. Column 1 shows the case, column 2 the baseline value with respect of the metric, column 3 the comparison case result, and column 4 the gain of the new solution compared with the baseline.

Table 3. Baseline Interplanetary Trajectory Performance

Case	Case (1): Baseline	Comparison	Gain
Case (2): Late escape date	2021-Oct-03 15:17:57.1 UTC	2021-Dec-16 05:21:30.9 UTC	73 days
Case (3): Max. Kinetic energy	1603 kJ	1709 kJ	103 kJ
Case (5): Min. Time of flight	368 days	288 days	80 days
Case (6): Earth swing-by	290 days (case (2))		3 days (case(2))
	530.17 kg	524.69 kg	-5.5 kg

Missed Thrust Analysis

A missed thrust robust trajectory becomes one of the extra priorities for missions utilizing SEP such as DART. It is essential to understand the robustness to missed thrust events in the trajectory design process of SEP missions for the mission's success and reliability.

The baseline is divided into points that will be used as missed thrust events. The selection of the points is every 14 days starting from the escape date. The two weeks step-size is selected to provide enough points to generate a trend line, as will be seen further. In principle any number of points can be selected as long as there is enough for a reasonable trend line. Therefore, for this analysis, the points selected are in time after the escape: 0, 14, 28, . . . , 252 days. Figure 8 shows the points on the reference trajectory. For each point, an optimal trajectory is found with the objective of initially coasting as much as possible and the resulting coast time compared with the baseline. The reference trajectory states are found by interpolation using a spline to obtain the points and all mission constraints were enforced. Note in Fig. 9, on the trajectory z-axis, the first thrust arc adjusts the spacecraft position for the flyby - outlined by the gray vertical line. The flyby happens on day

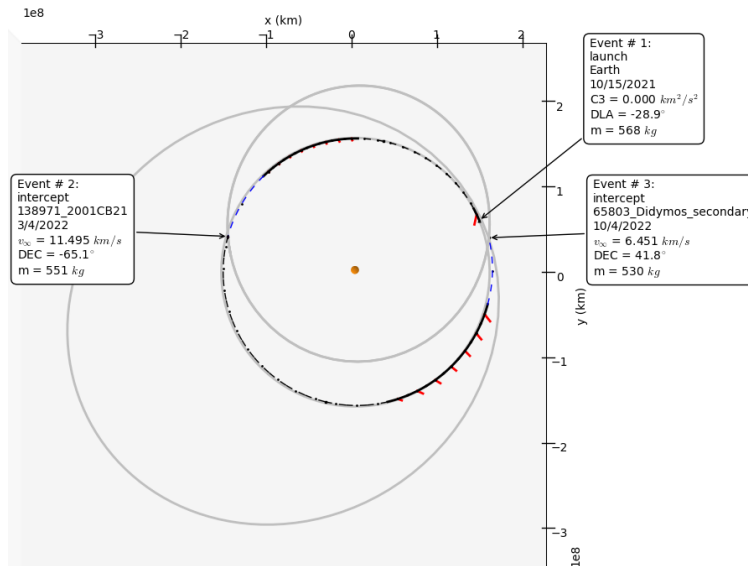


Figure 5. Case (4): Optimal trajectory that has the minimum time of flight with a fix escape date

152 and solutions prior to this date can leverage coast time if this flyby is not included.

The missed thrust is considered an emergency scenario, therefore the duty cycle can be increase to 95%. In order to get representative results for the overall mission, different fuel margins were considered in the analysis. A 0% fuel margin represents mass achieved in case (1) (530.1718 kg) and the other values are increments of Xe mass: 0% (0.0 kg), 1% (1.09 kg), 3% (3.29 kg), 5% (5.49 kg) and 7% (7.69 kg). The delivered dry mass will vary in accordance to the extra amount of propellant spent. Figure 10 shows the missed thrust analysis solution for a final condition with exactly the same spacecraft states as the baseline.

The comparison with the time in the coast arc is a good measure to check if the solution is indeed performing better, equal to or worse than the baseline. Note how the results with more fuel margin perform better, the increase in margin means that more fuel is available for the mission; therefore, the coast can be increased. As expected, there is an increase in performance during the coast arcs, with a pronounced dip after the flyby where a thrust arc is present. As the spacecraft approaches the impact, less time is available for maneuvering and the performance decreases. The results show that in almost all cases the mission can be recovered above the 14-day margin, except for the 0% margin between 252 to 294 days.

Improvements in the time taken for the mission to recover can be achieved by re-targeting the impact from the missed thrust event, as opposed to trying to reach the impact with exactly the same conditions. Two scenarios can be evaluated in this context: the asteroid impact is re-target respecting all the constraints, and the asteroid impact is re-target with some of the constraints relaxed. Table 4 presents the constraint values for the aforementioned options and Fig. 11 the optimization results.

As expected, the standard constraint results perform better than or equal to the fix constraints, and the results with more fuel margin are at least equal to results with higher dry mass. The relaxed constraints completely clear the 14-day margin for all the fuel margins throughout the trajectory. Figures 12 and 13 present the comparison between the three missed impact cases.

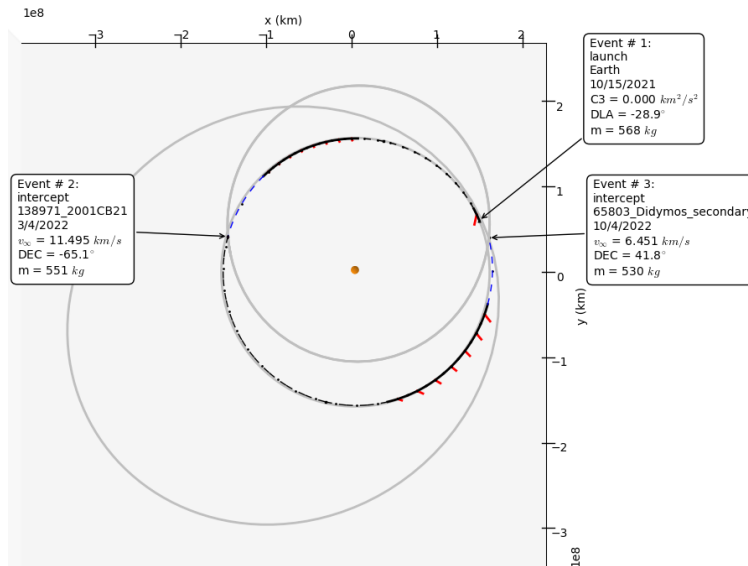


Figure 6. Case (5): Optimal trajectory that has the minimum time of flight without fixed dates

Table 4. Constraint Scenarios

Optimization Constraints	In-plane angle	Out-of-plane angle	Max. solar phase angle	Impact dates
Standard constraints	$175^\circ \leq \phi_{IP} \leq 180^\circ$	$-30^\circ \leq \phi_{OP} \leq 30^\circ$	60°	$\text{Sep. 25, 2022} \leq t_{\text{impact}} \leq \text{Oct. 20, 2022}$
Relaxed constraints	$170^\circ \leq \phi_{IP} \leq 180^\circ$	$-32.5^\circ \leq \phi_{OP} \leq 32.5^\circ$	75°	$\text{Sep. 25, 2022} \leq t_{\text{impact}} \leq \text{Oct. 20, 2022}$

Although it is clear that improvements can be achieved by a longer search, the values already obtained show a good margin. Some of the 0% margin results for the standard constraint are still below the 14-day margin near the end of the trajectory, but there is a clear improvement in the beginning of the trajectory, which is less critical compared with the non-fixed final conditions. The most important improvement in the trajectory's missed thrust is with the relaxed constraints near the end of the mission, where the fixed case presented recoveries with less than 14 days. Overall, mission success is guaranteed with a fuel margin of 5% for a 14-day period of engine failure at any point in the mission. Relaxing the constraints increase the mission robustness to missed thrust events, however, 1% of the allocated mass is already sufficient for the established goal of 14 days. In saving fuel mass on missed thrust, more mass can be allocated to other sub-systems or can result in larger margins.

ALTERNATIVE SCENARIOS

It was shown in the previous section that the baseline is robust to missed thrust. Although very promising, the results for the 0% fuel margin after 238 days cannot be recovered after a missed thrust event of 14 days. Day 238 can still recover in 14.71 days. In those cases, if the missed thrust is not detected in time for the proper corrections with a higher fuel margin the spacecraft will miss its impact with Didymos. A second missed impact probability is related to the poor knowledge of Didymos's orbit, where the spacecraft may arrive at the required final conditions but the target is not at the calculated position. Both scenarios need to be addressed and contingency plans need to

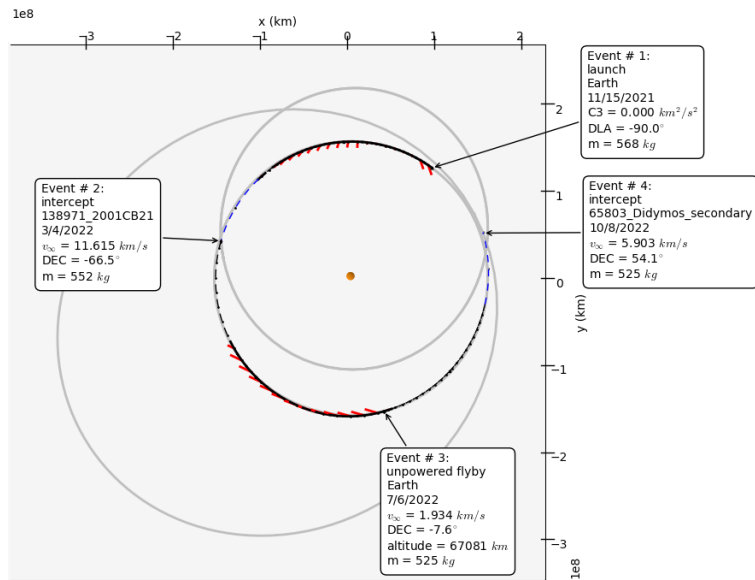


Figure 7. Case (6): Optimal trajectory with Earth gravity assist

be designed.

Go-Around Scenario

The go-around scenario targets an impact solution after 238 days. It was already shown that the spacecraft will not be able to hit the target in the initial time frame. Therefore, this solution targets the same constraints, except the arrival date, which leaves the spacecraft free to make one or more revolutions around the Sun. The duty-cycle is set to 90% and the optimizer is set for the maximum final mass in order to calculate the best possible Xenon margin.

Results presented in Fig. 14 show that the mission can reach the target roughly 2 years after the initial impact scenario. The obtained final mass translates to a propellant margin of 3.1%, which is still inside the allocated missed thrust mass budget. As done for the baseline, this scenario explored the use of an Earth swing-by to decrease the propellant consumption. The go-around trajectory with a swing-by (Fig. 15) consumes 16.26% of the propellant margin and, therefore, is not selected.

Missed Impact Scenario

This analysis is performed for the case where the spacecraft reaches the Didymos system correctly, but fails to impact Didymoon. Trajectory design for the missed impact starts on the final date and states of the baseline and, similarly to what was done before, optimizes the final mass using all constraints except arrival date. This direct re-targeting results in a final spacecraft mass well below the allocated margins, Fig. 16.

Although a direct re-targeting is not feasible, the former go-around scenario points to the fact that a solution for the missed impact can be found with a small change in the reference trajectory. Also, the missed thrust analysis shows a comfortable Xenon margin, which can be used to design a new baseline with a slightly higher propellant consumption. The new baseline still includes the flyby (138971) 2001 CB₂₁ and two Didymoon encounters with all the constraints adopted except

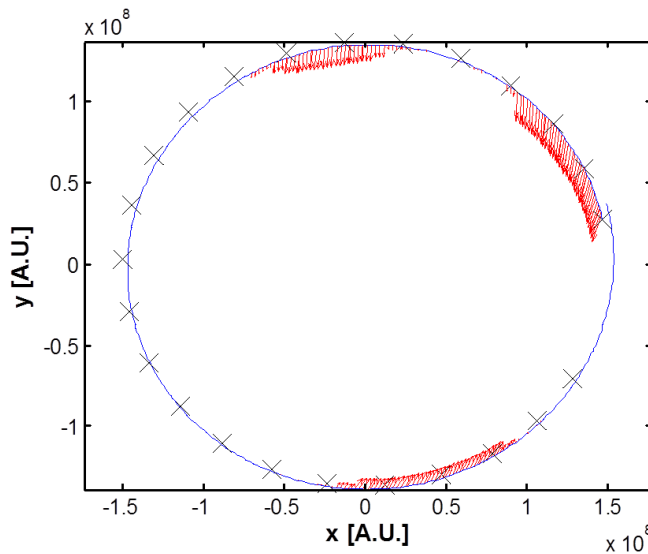


Figure 8. Missed thrust events

the arrival dates for the second Didymos encounter. The solution is presented in Fig. 17 and results in a propellant margin of around 3%, which is still inside the allocated margin. The spacecraft is put in a resonant trajectory with Didymos after the first encounter, which results in an almost ballistic Didymos-to-Didymos trajectory.

CONCLUSIONS

The NASA Double Asteroid Redirection Test will be the first mission to test the kinetic impactor concept for planetary defense. The mission targets the impact of the secondary body of the (65803) Didymos system in conditions that make the orbit change (around the primary) measurable from Earth. The Heliocentric transfer was designed using an optimization tool that includes the mission constraints for a maximum final spacecraft mass. This solution was incorporated into the mission's Heliocentric baseline and is composed of a thrust-coast-thrust structure, making it robust to impact condition changes. It was shown that DART's main trajectory has good performance when compared with the other objectives and the missed thrust analysis showed a robust trajectory against engine failure or other safe-mode event. Due to the trajectory's profile, the flexibility in re-adjusting the trajectory is considerable. If the engine failure happens in the first arc, there is enough time to redirect the trajectory to the main target (midcourse flyby is discarded). If the missed thrust happens in the more critical second arc, less time is available for target redirection considering that the same amount of fuel is available for maneuvering. However, the trajectory already has the necessary energy and critical angles remaining, so only small adjustments to the velocity vector and the timing for the impact angle's in-plane component are required for the trajectory to recover. Results show that by increasing the engine duty cycle to 95%, the trajectory can be redirected with the same amount of Xenon until the beginning of the second arc for failures up to 14 days. For missed thrust on the final arc, the mission can be redirected with an additional 1% Xenon margin, 1.09 kg. Lastly, a considerable improvement on the robustness can be achieved by targeting the normal and relaxed constraints rather than fixing the impact conditions.

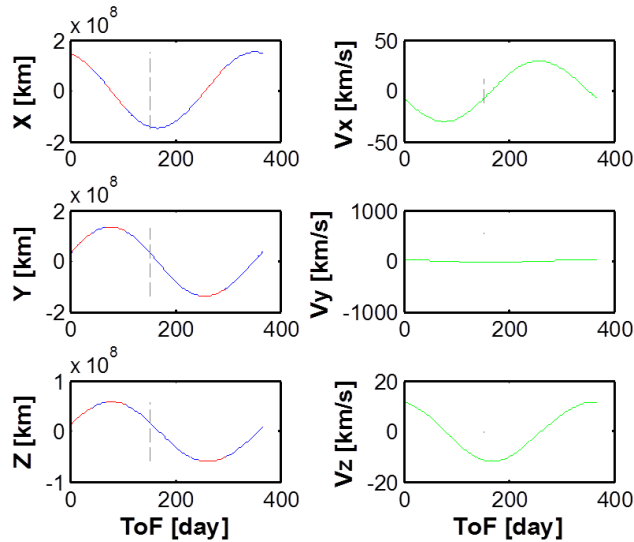


Figure 9. Baseline trajectory position and velocity in time

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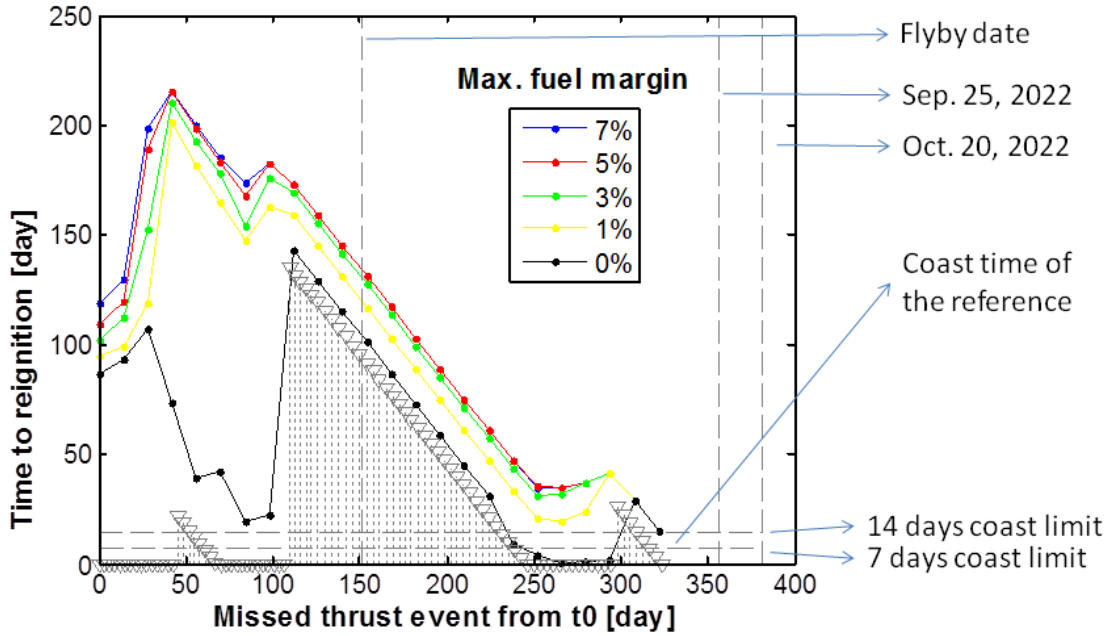


Figure 10. Missed thrust analysis for fix impact conditions

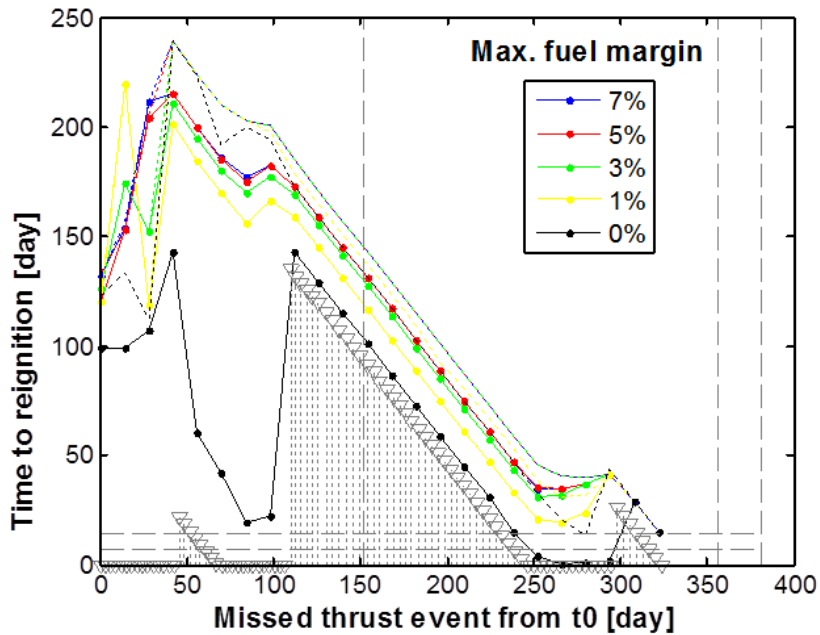


Figure 11. Missed thrust analysis for normal and relaxed impact conditions (solid line normal constrains, dotted line relaxed constraints)

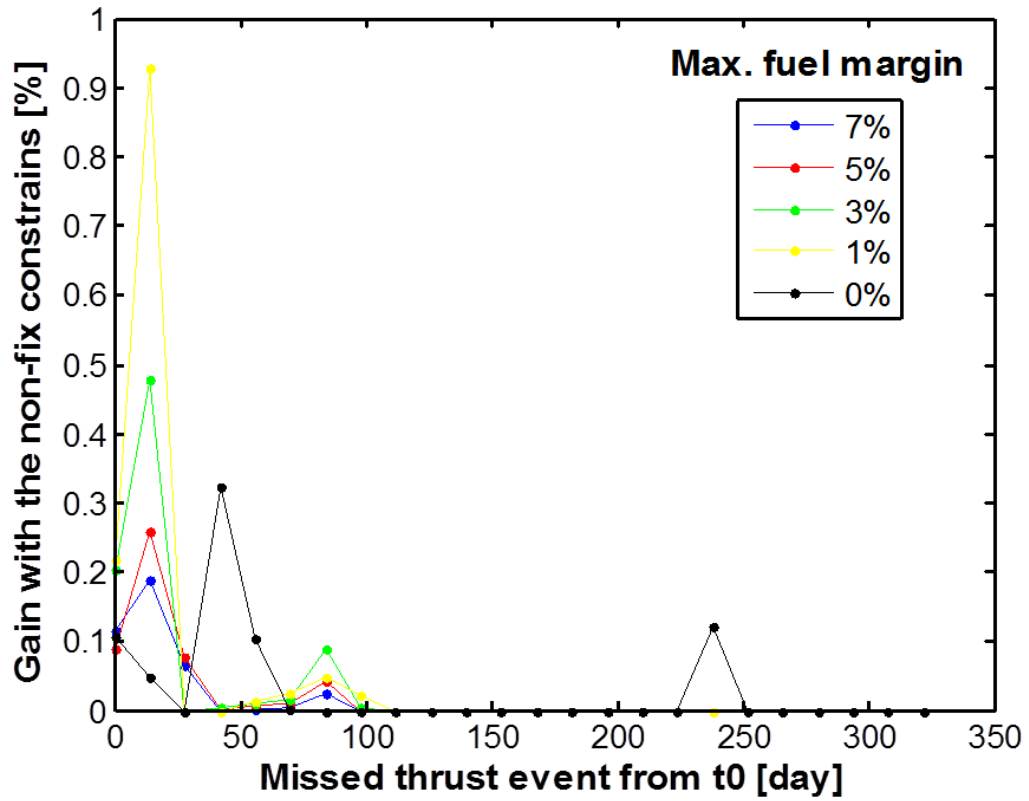


Figure 12. Missed thrust Fix Vs. Standard comparison

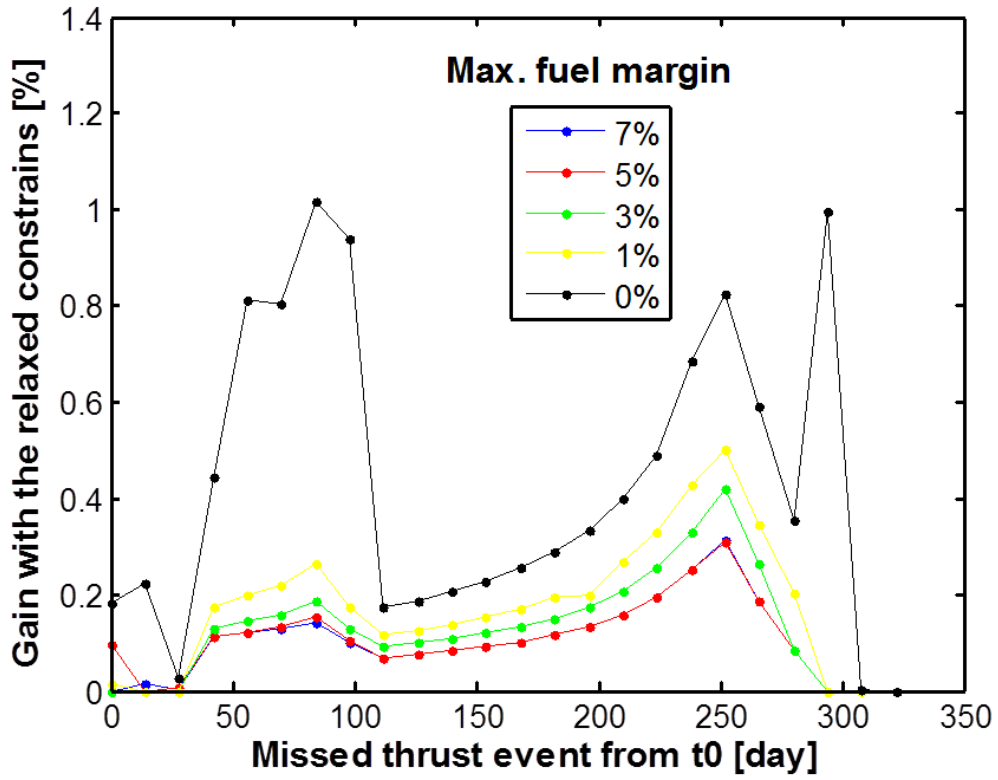


Figure 13. Missed thrust Standard Vs. Relaxed comparison

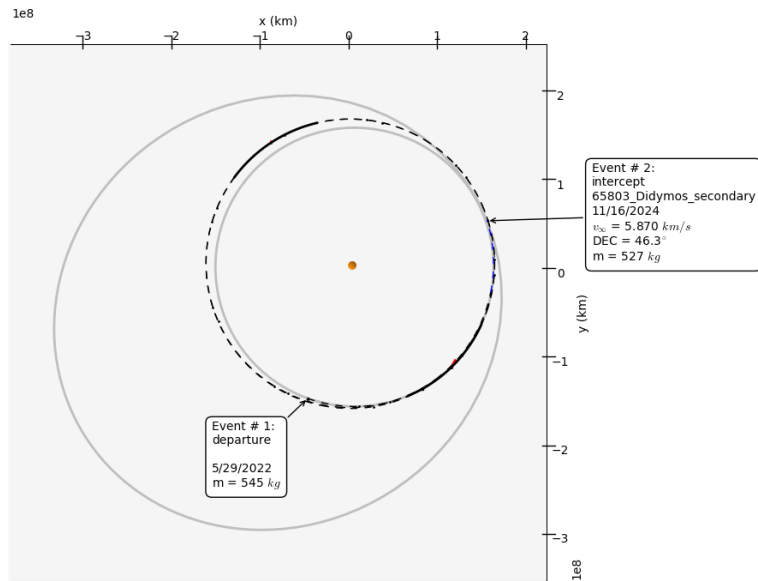


Figure 14. Go-around Scenario

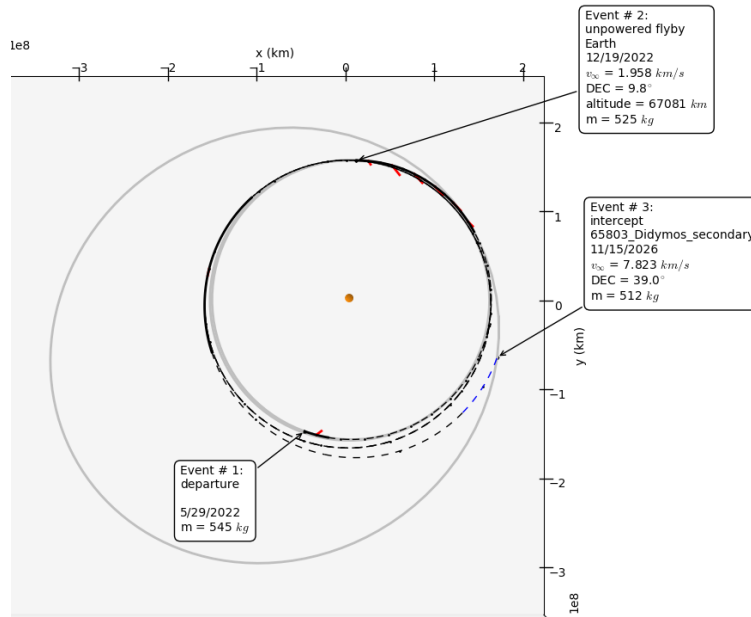


Figure 15. Go-around Scenario with Earth swing-by

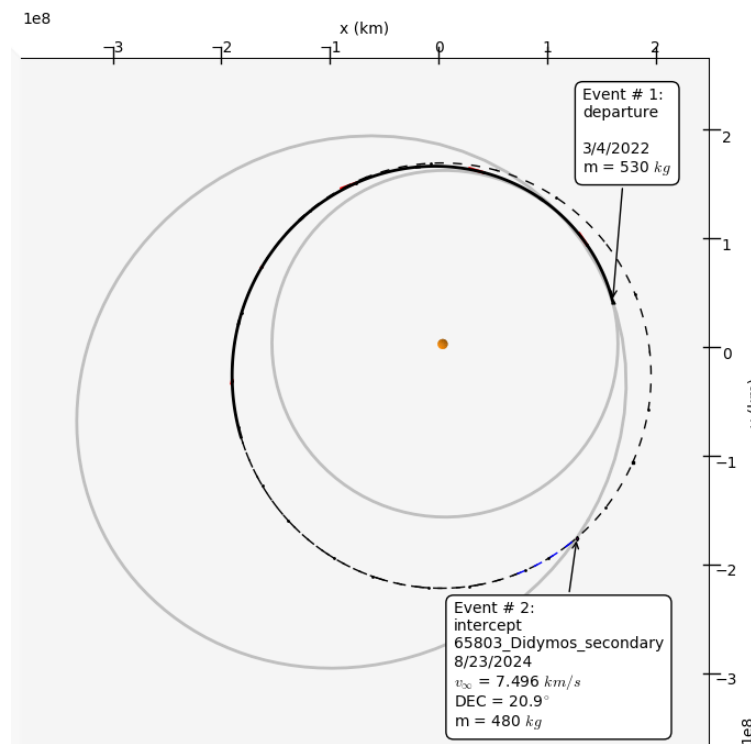


Figure 16. Missed impact Scenario with direct re-targeting

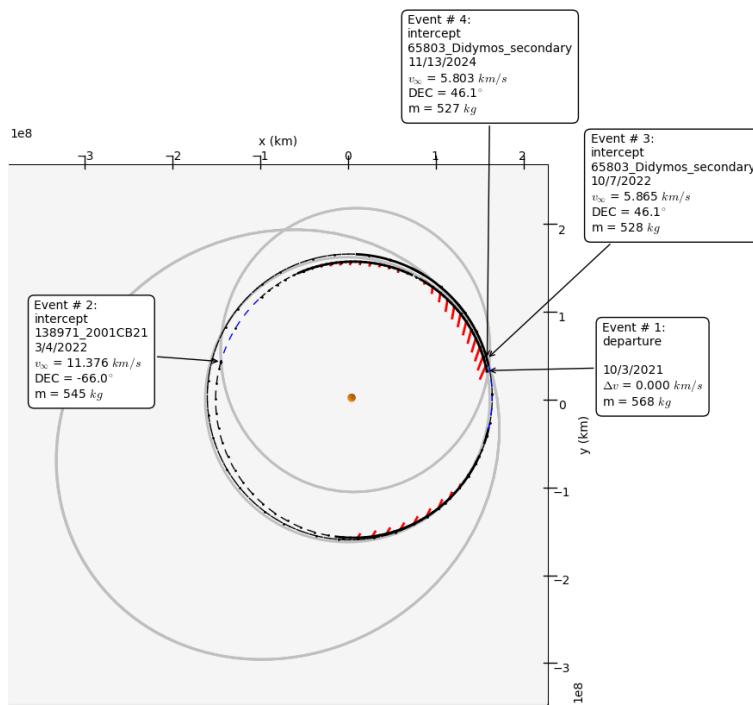


Figure 17. Missed impact Scenario with two encounters